APPENDIX A

Airfoil Pressure Distributions

The aerodynamic performance of airfoil sections can be studied most easily by reference to the distribution of pressure over the airfoil. This distribution is usually expressed in terms of the pressure coefficient:

$$c_p = \frac{p - p_\infty}{\frac{1}{2}\rho U_\infty^2}$$

Cp is the difference between local static pressure and freestream static pressure, nondimensionalized by the freestream dynamic pressure.

What does an airfoil pressure distribution look like? We generally plot C_p vs. x/c.

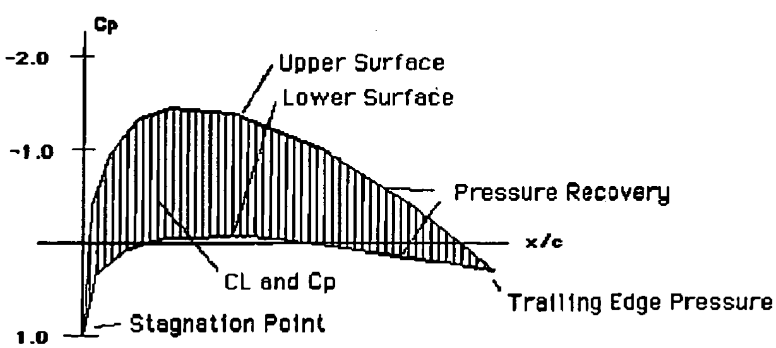
x/c varies from 0 at the leading edge to 1.0 at the trailing edge. C_p is plotted "upside-down" with negative values (suction), higher on the plot. (This is done so that the upper surface of a conventional lifting airfoil corresponds to the upper curve.)

The C_p starts from about 1.0 at the stagnation point near the leading edge...

It rises rapidly (pressure decreases) on both the upper and lower surfaces...

...and finally recovers to a small positive value of C_p near the trailing edge.

Various parts of the pressure distribution are depicted in the figure below and are described in the following sections.



- Upper Surface
  The upper surface pressure is lower (plotted higher on the usual scale) than the lower surface Cp in this case. But it doesn't have to be.

- Lower Surface

The lower surface sometimes carries a positive pressure, but at many design conditions is actually pulling the wing downward. In this case, some suction (negative Cp -> downward force on lower surface) is present near the midchord.

- Pressure Recovery
  This region of the pressure distribution is called the pressure recovery region.
  The pressure increases from its minimum value to the value at the trailing edge.
  This area is also known as the region of adverse pressure gradient. As discussed in other sections, the adverse pressure gradient is associated with boundary layer transition and possibly separation, if the gradient is too severe.

- Trailing Edge Pressure
  The pressure at the trailing edge is related to the airfoil thickness and shape near the trailing edge. For thick airfoils the pressure here is slightly positive (the velocity is a bit less than the freestream velocity). For infinitely thin sections $C_p = 0$ at the trailing edge. Large positive values of C_p at the trailing edge imply more severe adverse pressure gradients.

- C_L and C_p
  The section lift coefficient is related to the C_p by: $C_l = \text{int}\ (C_{pl} - C_{pu})\ dx/c$
  (It is the area between the curves.)
  with C_{pu} = upper surface C_p
  and recall C_l = section lift / (q c)

- Stagnation Point
  The stagnation point occurs near the leading edge. It is the place at which V = 0. Note that in incompressible flow $C_p = 1.0$ at this point. In compressible flow it may be somewhat larger.

We can get a more intuitive picture of the pressure distribution by looking at some examples and this is done in some of the following sections in this chapter.

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